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# NACA

# RESEARCH MEMORANDUM

## AN INVESTIGATION OF A SUPERSONIC AIRCRAFT CONFIGURATION HAVING A TAPERED WING WITH CIRCULAR-ARC SECTIONS AND 40° SWEEPBACK

## PRESSURE-DISTRIBUTION MEASUREMENTS OF THE INTERFERENCE EFFECT OF THE WING ON THE FUSELAGE AT MACH NUMBERS OF 1.40 AND 1.59


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RESEARCH MEMORANDUM

# AN INVESTIGATION OF A SUPERSONIC AIRCRAFT CONFIGURATION HAVING A TAPERED WING WITH CIRCULAR-ARC SECTIONS AND 40° SWEEPBACK

## PRESSURE-DISTRIBUTION MEASUREMENTS OF THE INTERFERENCE EFFECT OF THE WING ON THE FUSELAGE AT MACH NUMBERS OF 1.40 AND 1.59

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## SUMMARY

The interference effect of the wing on the fuselage of a supersonic aircraft configuration having a tapered sweptback wing with 3° incidence has been determined from pressure measurements obtained during an investigation of this configuration in the Langley 4- by 4-foot supersonic tunnel. Tests were conducted at Mach numbers of 1.40 and 1.59, and at a Reynolds number based on fuselage length of approximately $2.7 \times 10^6$.

The pressure measurements for this configuration showed that the wing-lift carry-over to the body was confined primarily to the part of the fuselage behind the wing trailing edge. For an angle-of-attack range from 0° to 8°, the integrated normal-force coefficient of the wing-fuselage combination was approximately 6 percent less than the value obtained from the experimental loading over the wing in the presence of the body when extrapolated from the 0.186-semispan station to the center line of the body. The effect of the fuselage was to decrease the estimated normal force carried by the inboard panel of the wing by approximately 45 percent at an angle of attack of 4° and 25 percent at an angle of attack of 8°.

## INTRODUCTION

A knowledge of the interference effects at supersonic speeds between the wing and the body of an aircraft configuration is important when the aerodynamic loads and characteristics of the wing-body combination are being evaluated. The problem is a complex one for which theoretical solutions have been obtained only for special configurations under idealized assumptions (see, for instance, refs. 1 to 3). Unfortunately, the amount of experimental data available is limited, particularly the results of pressure-distribution studies.

The purpose of this paper is to present pressure measurements illustrating the interference effect of a wing on a body of a specific supersonic aircraft configuration. These results were obtained during an investigation of this configuration in the Langley 4- by 4-foot supersonic tunnel at Mach numbers of 1.40 and 1.59 and a Reynolds number based on fuselage length of approximately $2.7 \times 10^6$. In addition, the effect of wing-tip skids on the pressure distribution over the wing is shown.

## SYMBOLS

$\rho$ mass density of air

$V$ airspeed

$a$ speed of sound in air

$M$ Mach number, $V/a$

$q$ dynamic pressure, $\rho V^2/2$

$p$ free-stream static pressure

$p_l$ local static pressure

$P$ pressure coefficient, $\dfrac{p_l - p}{q}$

$y$ coordinate measured perpendicular to plane of symmetry of fuselage

$b$ wing span

$\alpha$ angle of attack of fuselage

$\varphi$ fuselage polar angle ($0^\circ$ at bottom)

$d$ diameter of fuselage at any point

$c$ airfoil chord or fuselage length at any spanwise station

$\bar{c}$ mean chord, $S/b$

$S$ wing area (wing extended through fuselage)

$c_n$ section normal-force coefficient

## MODEL

The model shown in figure 1 had a $40^\circ$ sweptback wing with $0^\circ$ twist, aspect ratio of 4, and taper ratio of 0.5. The wing was at an incidence angle of $3^\circ$ relative to the body axis. The airfoil sections in planes perpendicular to the quarter-chord line were symmetrical circular arcs with thickness ratios of 10 percent. The aerodynamic characteristics of the wing are given in reference 4 for a Mach number of 1.40 and in reference 5 for a Mach number of 1.59. The model was sting-mounted in the tunnel as shown in figure 2.

The basic fuselage was a body of revolution with a length of 30.267 inches, a fineness ratio of 9.4, and a ratio of wing span to maximum body diameter of 8.04. The wing was mounted low on the fuselage as shown on figure 1. The top and bottom fuselage canopies were removable for testing as a body of revolution. The fuselage coordinates are given in reference 6.

Pressure orifices were located at nine stations along the body at the six radial positions shown in figure 1. Additional orifices were located along the top ($\phi = 180^\circ$) of the fuselage but could be used only when the top canopy was removed. Pressure orifices were also located along the top surface fillet of the wing-body juncture.

Tip skids were installed on the wing as shown in figure 1.

## TESTS

The tests were conducted in the Langley 4- by 4-foot supersonic tunnel at Mach numbers of 1.40 and 1.59. A detailed description of the tunnel, as well as the calibration data of the test section for a Mach number of 1.59, is presented in reference 6. The calibration data of the test section for a Mach number of 1.40 are presented in reference 7.

The tests were conducted under tunnel stagnation conditions of: pressure, 0.25 atmosphere; temperature, 110° F; and dew point, -35° F. The calibration data indicated negligible condensation effects at these conditions.

Pressure measurements over the fuselage were recorded for the condition of the fuselage alone and also for that of the wing-fuselage combination. Two different fuselage configurations were used. The fuselage was tested as a body of revolution (canopies and tail removed) at $M = 1.59$. In addition, the fuselage was tested at $M = 1.40$ and $M = 1.59$ as a body of revolution with top and bottom canopies and tail assembly.

Since the bottom fuselage canopy was integral with the wing, a canopy effect was involved in the comparison between the pressure distributions with and without the wing for those tests designated as body-of-revolution tests. This effect was small, however, and has been neglected in the comparison.

The angle-of-attack range was from -5° to 8°. The Reynolds numbers for these tests, based on fuselage length, were $2.60 \times 10^6$ for a Mach number of 1.59 and $2.70 \times 10^6$ for a Mach number of 1.40.

## ACCURACY

Since the magnitude of the free-stream flow angle, Mach number, and pressure gradients are small in the vicinity of the model, no corrections due to these sources have been applied to the data. It is estimated that the accuracy of the data is as follows:

| | |
|---|---|
| Mach number | ±0.01 |
| Angle of attack, deg | |
| Geometric measurement (probable error) | ±0.02 |
| Maximum flow irregularity | ±0.10 |
| Absolute pressure coefficient | ±0.010 |

## RESULTS AND DISCUSSION

Wing-body interference.- Comparisons of the longitudinal pressure distributions over the fuselage, with and without the wing, at a Mach number of 1.59 are shown in figure 3 for a range of angles of attack of from $-5^\circ$ to $8^\circ$. The wing was at an incidence angle of $3^\circ$. For these tests, the fuselage configuration was a body of revolution. A sketch of the model is shown in each figure, together with a representation of the theoretical forward limit of the region of wing influence on the body. This limit is defined on the body by two opposing helices originating at the wing leading-edge fuselage junctures with the helix angle equal to the Mach angle.

Also shown in these figures are the linearized theoretical pressure distributions for the body alone (refs. 8 and 9) and the experimental chordwise pressure distributions over the upper and lower surfaces of the inboard station of the wing $\left(y = 0.186\,\frac{b}{2}\right)$. The pressure variations over this wing station are presented so that an indication of the relationship between wing pressure and the change in fuselage pressure due to the wing may be obtained.

For positive angles of attack, the wing caused a decrease in the pressures over the upper surface of the rear part of the body and an increase in the pressures over the lower surface so that an increase in the normal-force coefficient of the fuselage results. The distribution of this increase may be seen in figure 4 where the normal-force loading distribution of the fuselage with and without the wing is shown plotted along the body axis.

The rearward shift in the center of pressure of the fuselage, due to the wing-lift carry-over, resulted in a change in the pitching-moment coefficient of the fuselage from a condition of instability (about the quarter-chord of the M.A.C.) to one of approximate neutral stability.

The fuselage normal-force loading distribution in the spanwise direction is shown in figure 5. Also shown in this figure are the experimental wing loading distributions (obtained from ref. 5). Since the normal-force distribution in the region of the wing-fuselage juncture is not known, that part of the curve is represented by a dashed line.

The integrated normal-force coefficient of the wing-fuselage combination was less than the normal-force coefficient obtained from the experimental loading over the wing in the presence of the body when extrapolated from the 0.186-semispan station to the center line of the body. This decrease in normal force amounted to approximately 7 percent for an angle of attack of $4^\circ$ and 5 percent for an angle of attack of $8^\circ$.

With respect to the estimated normal force carried by the inboard panel of the wing (extrapolated value), the effect of the fuselage was to decrease this normal force by approximately 45 percent at an angle of attack of $4^{\circ}$ and 25 percent at an angle of attack of $8^{\circ}$.

A comparison of the pressure distributions over the model fuselage with and without the wing for the case of the fuselage with attached canopies and tail assembly is presented in figure 6 for Mach numbers of 1.40 and 1.59. The flagged symbols denote values at a Mach number of 1.40. Although the results are not so complete at $\phi = 180^{\circ}$ as for the case of the fuselage without canopies, the same general effects of the wing on the fuselage pressures are shown. The effect of the tail assembly was not apparent because of the lack of data in this region.

The pressure distribution on the fuselage in the immediate vicinity of the wing-body juncture is presented in figure 7(a) for a Mach number of 1.40, and in figure 7(b) for a Mach number of 1.59. In general, the pressure distribution at this station is similar to that over the upper surface of the inboard station of the wing $\left(y = 0.186 \frac{b}{2}\right)$.

Wing-tip skid interference.- The effect of tip skids on the pressure distribution over the lower surface of the wing in the vicinity of the tip $\left(y = 0.937 \frac{b}{2}\right)$ is shown in figure 8. The addition of tip skids caused an increase in the pressures over the rear part of the lower surface of the wing in the region of the tip. No effect was noted on the pressure distribution over the upper surface of the wing. This increase in pressure resulted in the addition of a slight stabilizing increment to the total pitching moment of the configuration.

## CONCLUDING REMARKS

The interference effect of the wing on the fuselage of a supersonic aircraft configuration having a tapered sweptback wing with $3^{\circ}$ incidence has been determined from pressure measurements obtained during an investigation of this configuration in the Langley 4- by 4-foot supersonic tunnel. Tests were conducted at Mach numbers of 1.40 and 1.59, and at a Reynolds number based on fuselage length of approximately $2.7 \times 10^{6}$.

The pressure measurements for this configuration showed that the wing-lift carry-over to the body was confined primarily to the part of the fuselage behind the wing trailing edge. For an angle-of-attack range from $0^{\circ}$ to $8^{\circ}$, the integrated normal-force coefficient of the wing-fuselage combination was approximately 6 percent less than the value obtained from the experimental loading over the wing in the presence of the body when extrapolated from the 0.186-semispan station to the center line of the body. The effect of the fuselage was to decrease the estimated normal force carried by the inboard panel of the wing by approximately 45 percent at an angle of attack of $4^{\circ}$ and 25 percent at an angle of attack of $8^{\circ}$.

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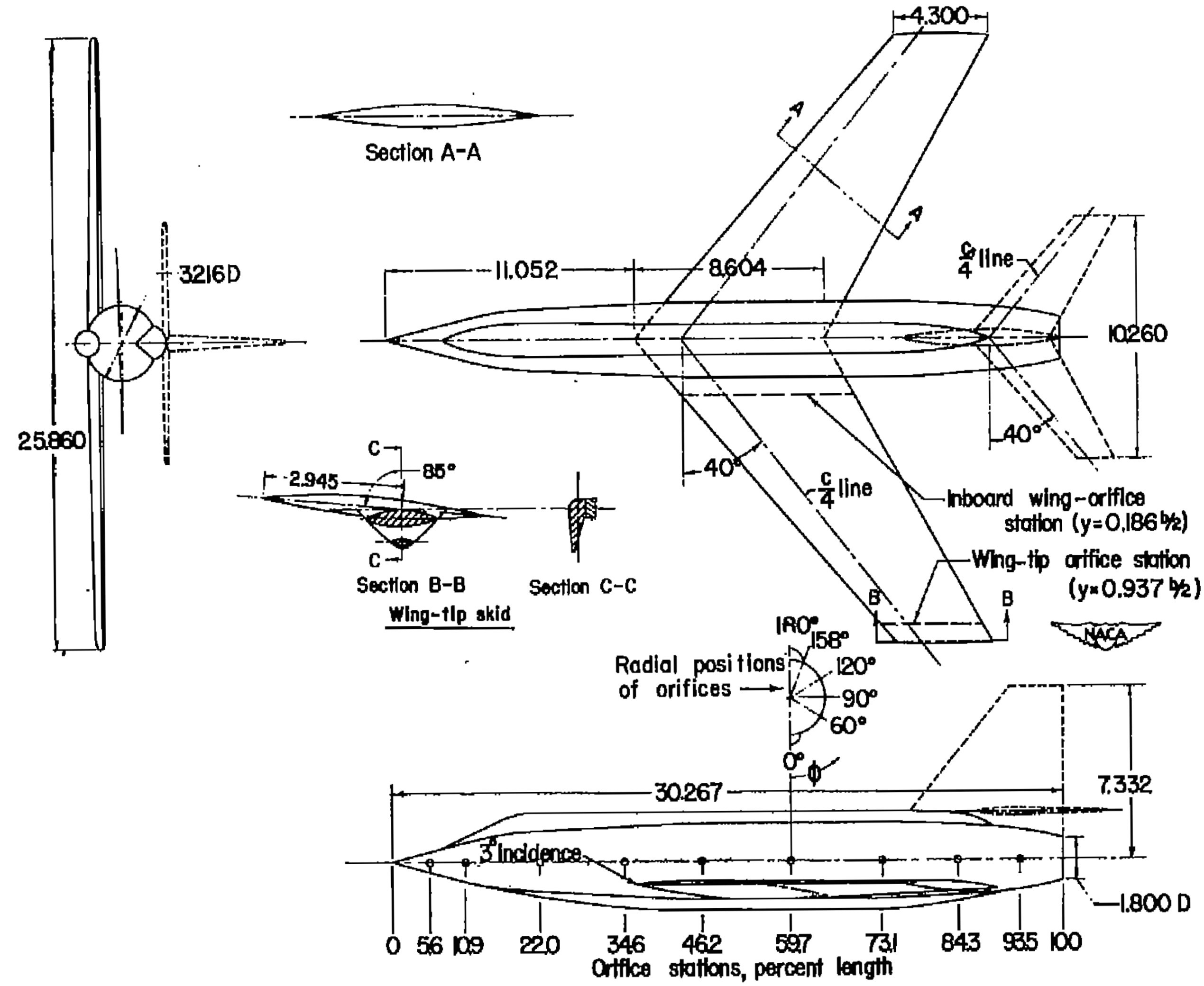
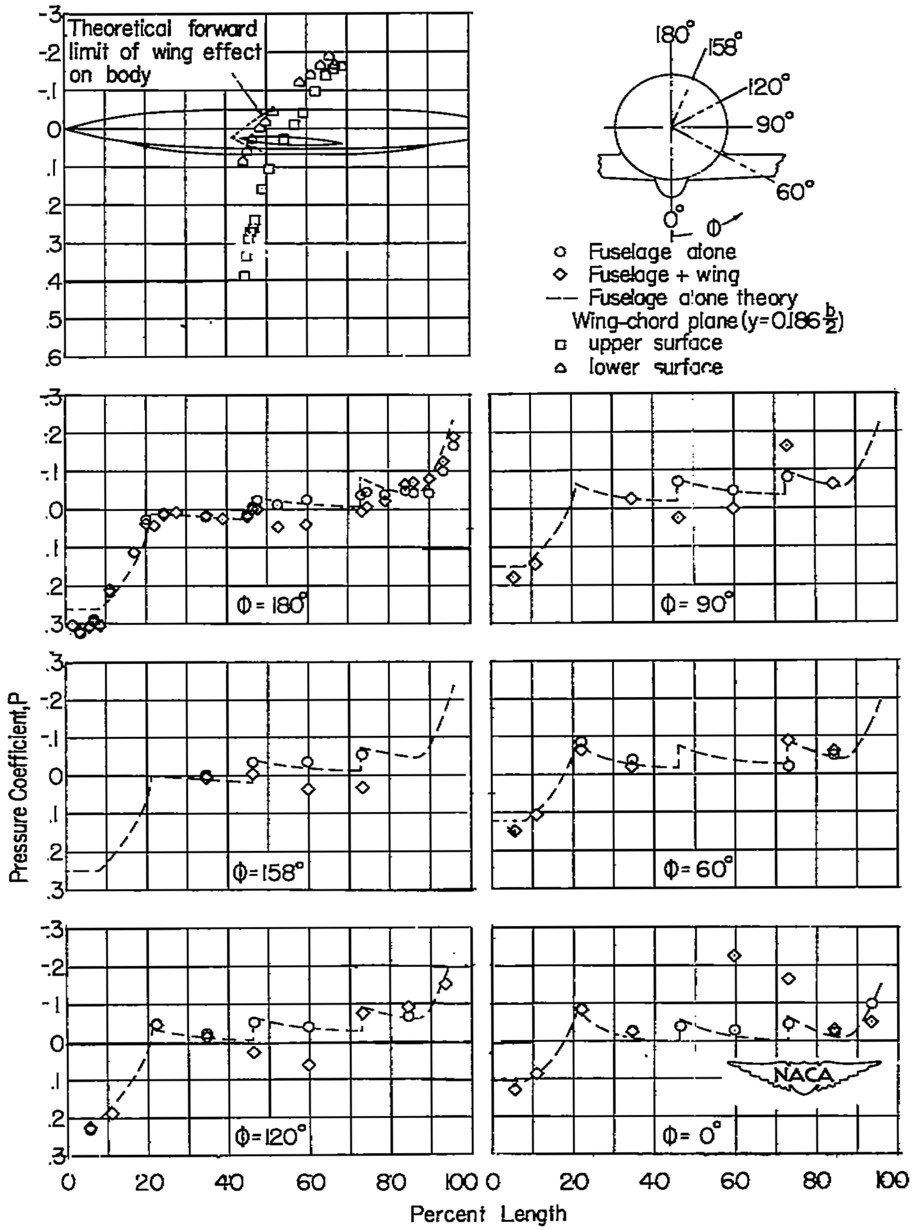


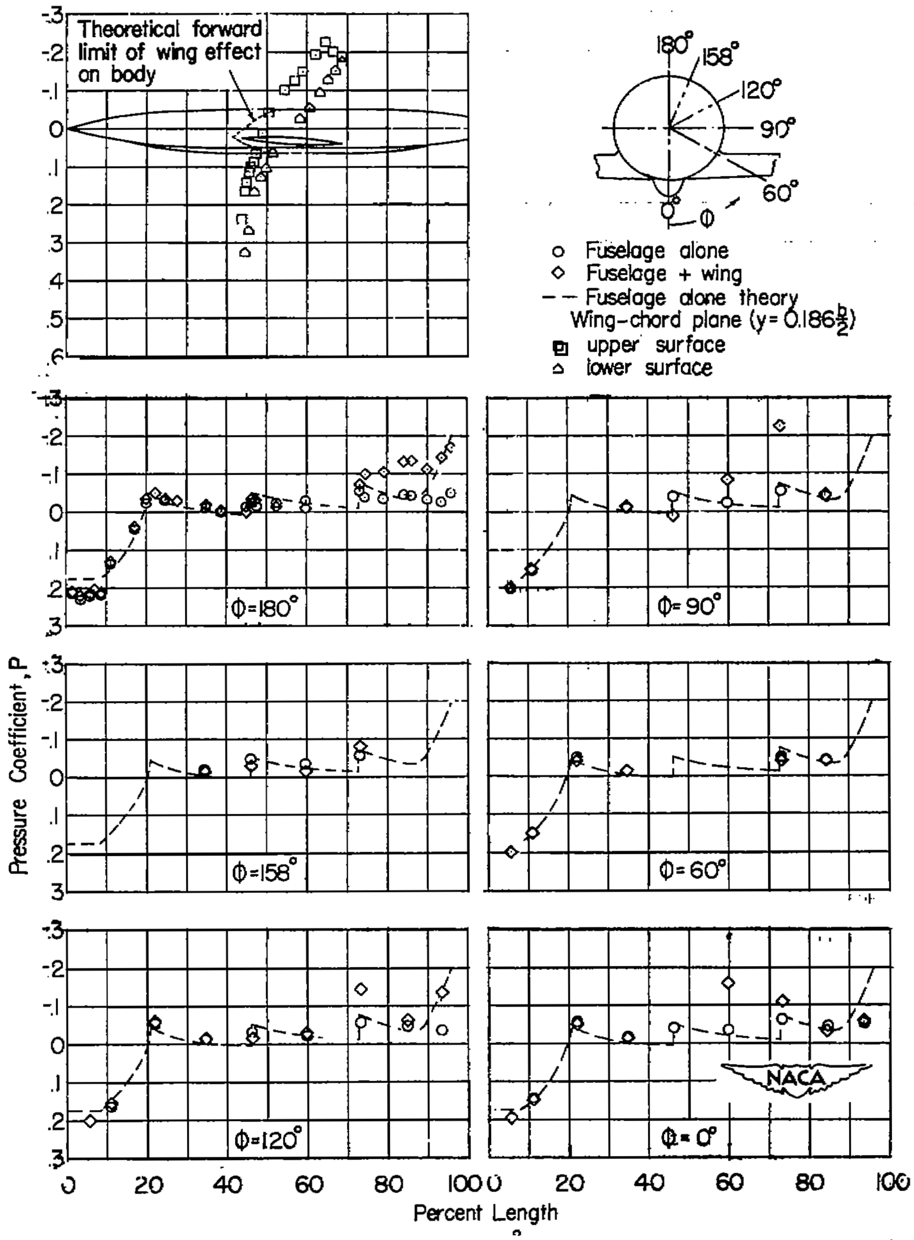

Figure 1.- Details of model of supersonic aircraft configuration. Dimensions are in inches unless otherwise noted.


Figure 2.- Downstream view of test model mounted in the Langley 4- by 4-foot supersonic tunnel.



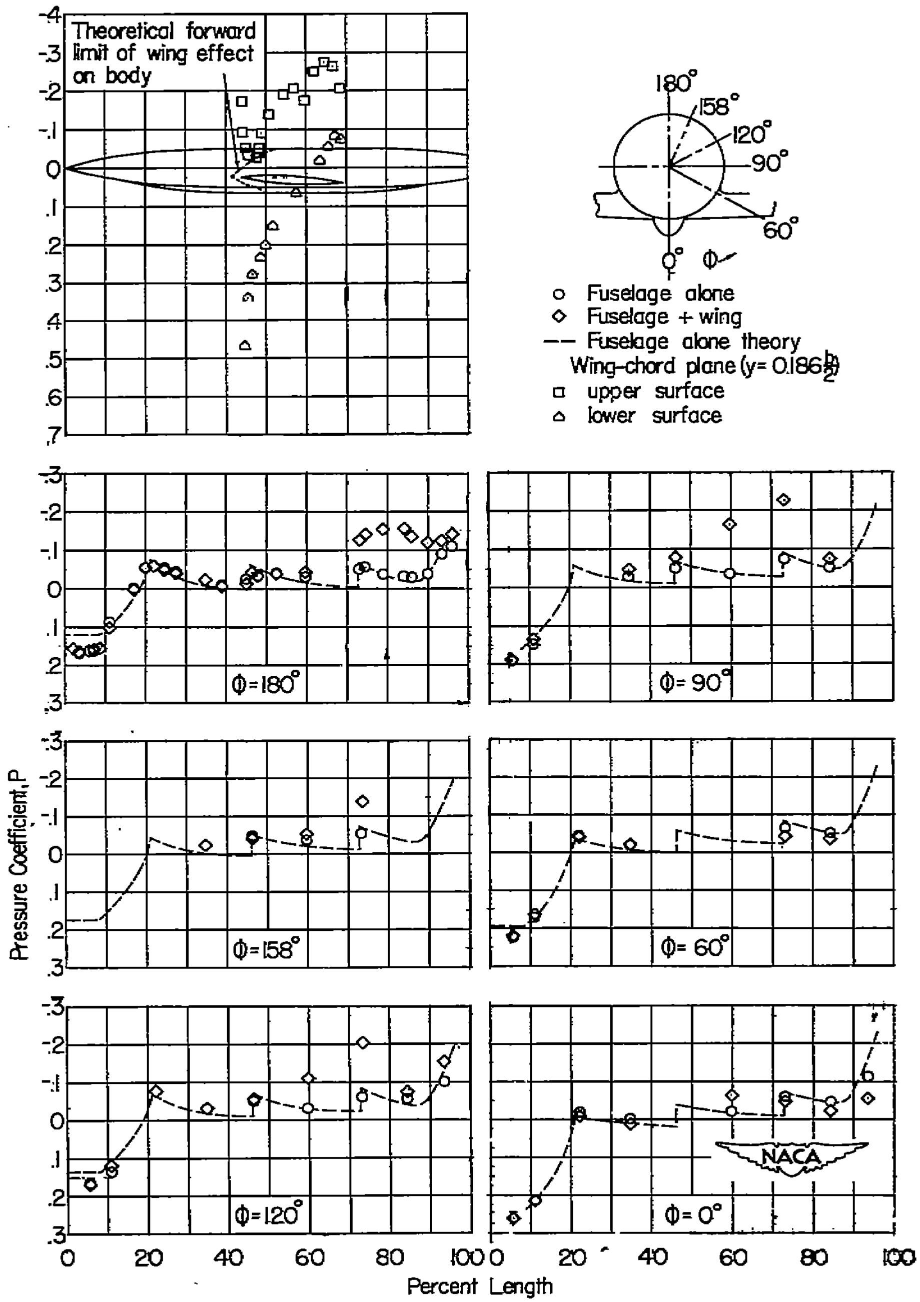

(a) $\alpha = -5^{\circ}$.

Figure 3.- Pressure distribution over the fuselage with and without the wing. M = 1.59.



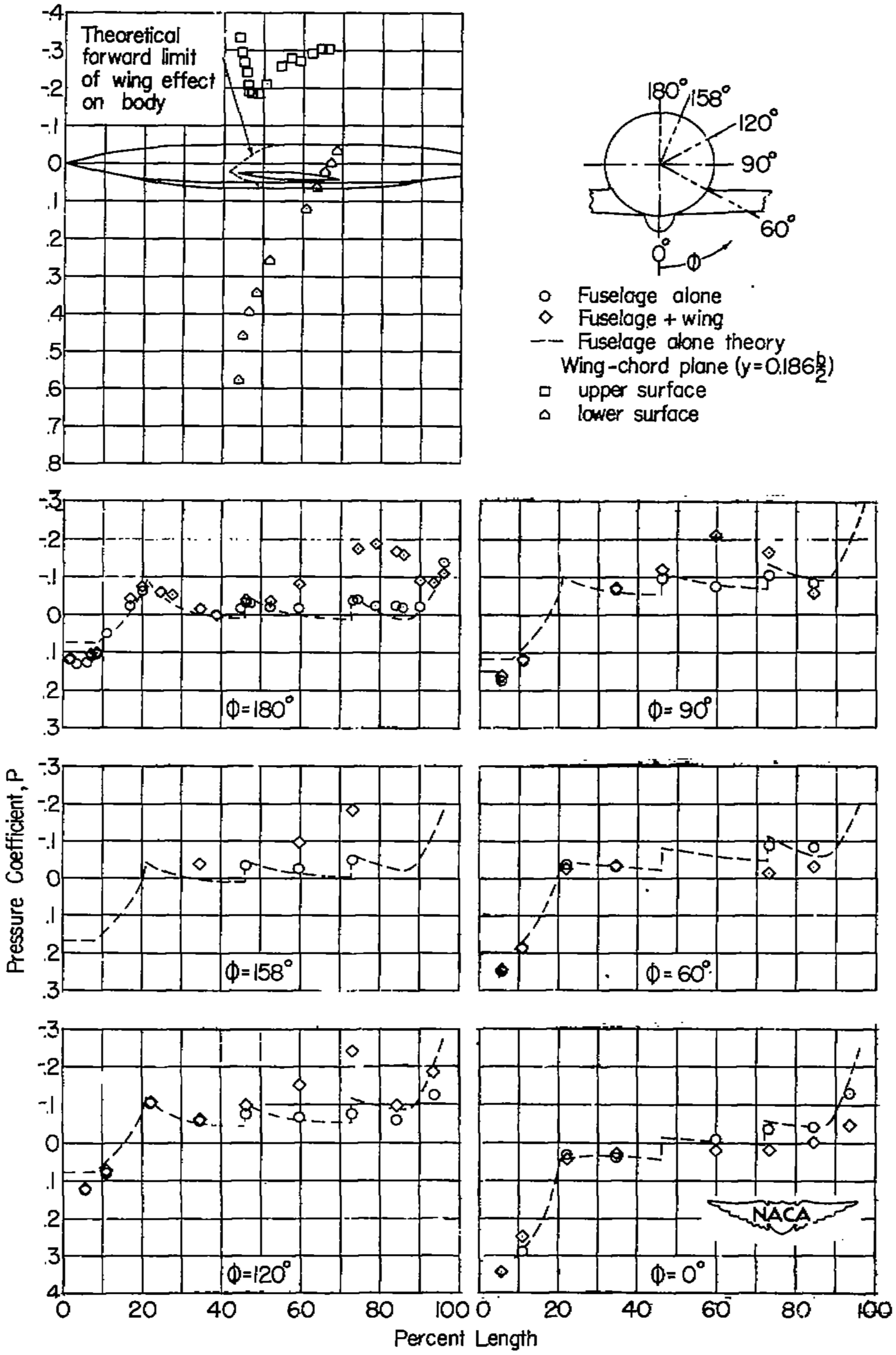

(b) $\alpha = 0^{\circ}$.

Figure 3.- Continued.




(c) $\alpha = 4^{\circ}$.

Figure 3.- Continued.




(d) $\alpha = 8^{\circ}$.

Figure 3.- Concluded.

[redacted]

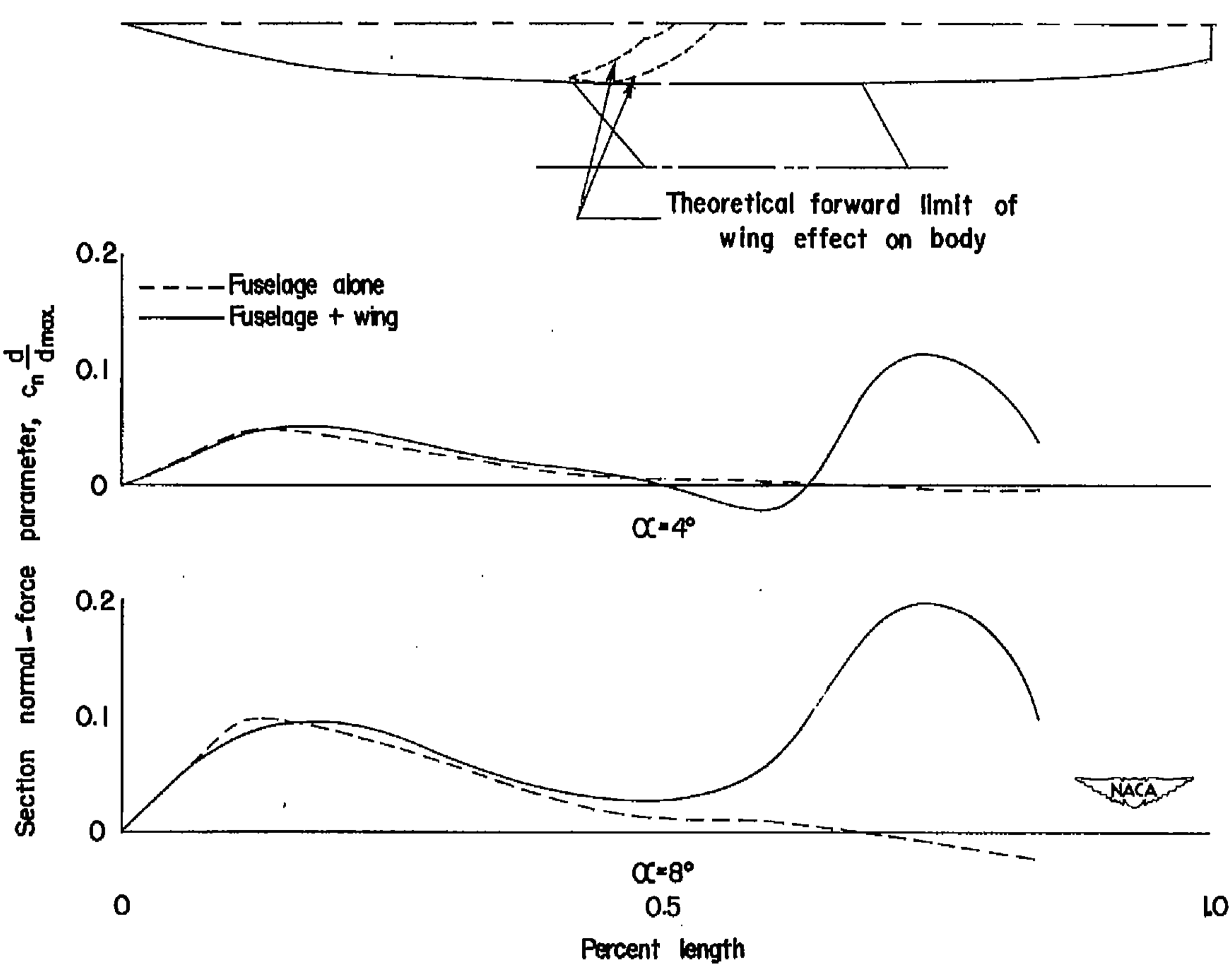



Figure 4.- Normal-force loading distribution as a function of body length. M = 1.59.

[redacted]

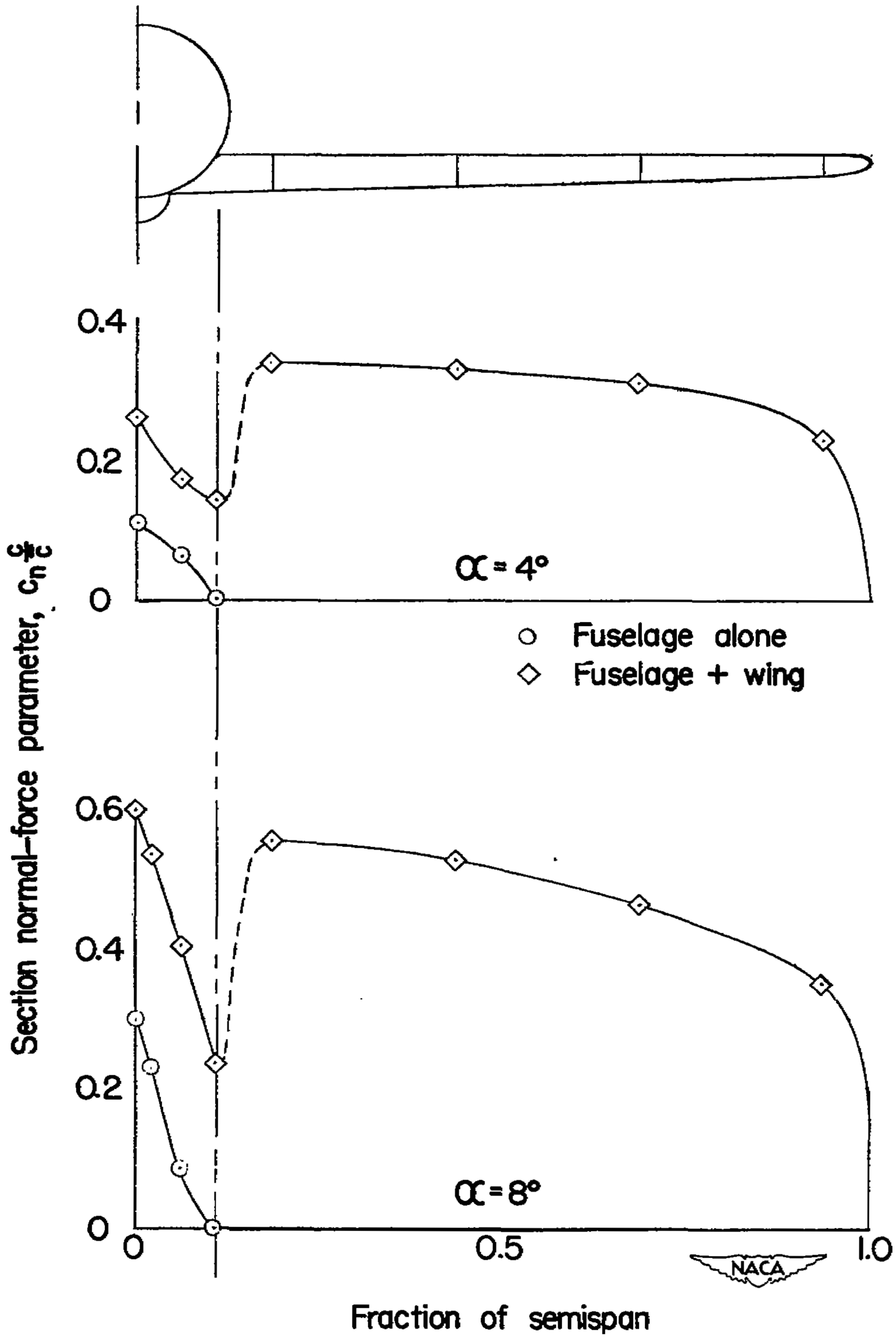
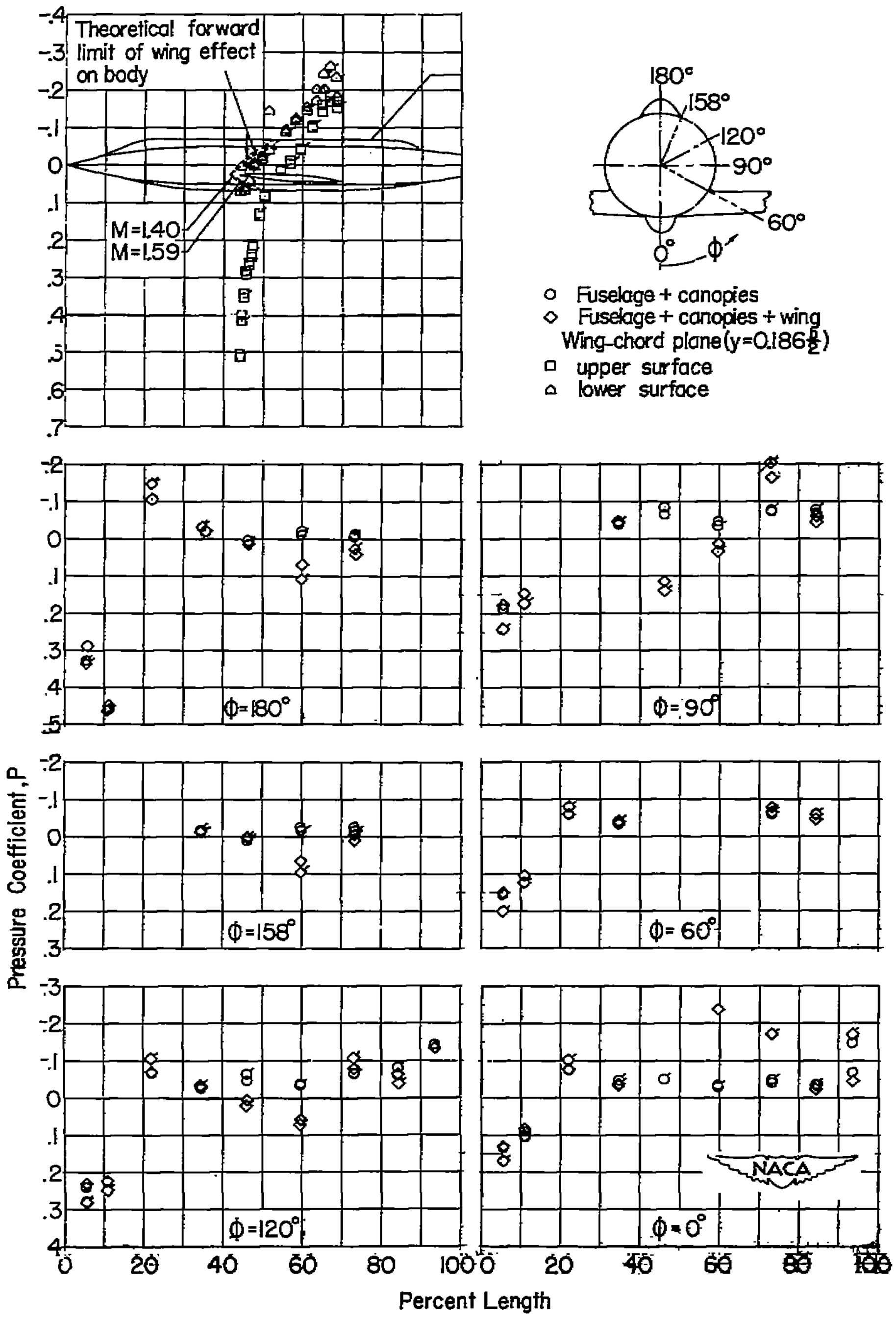
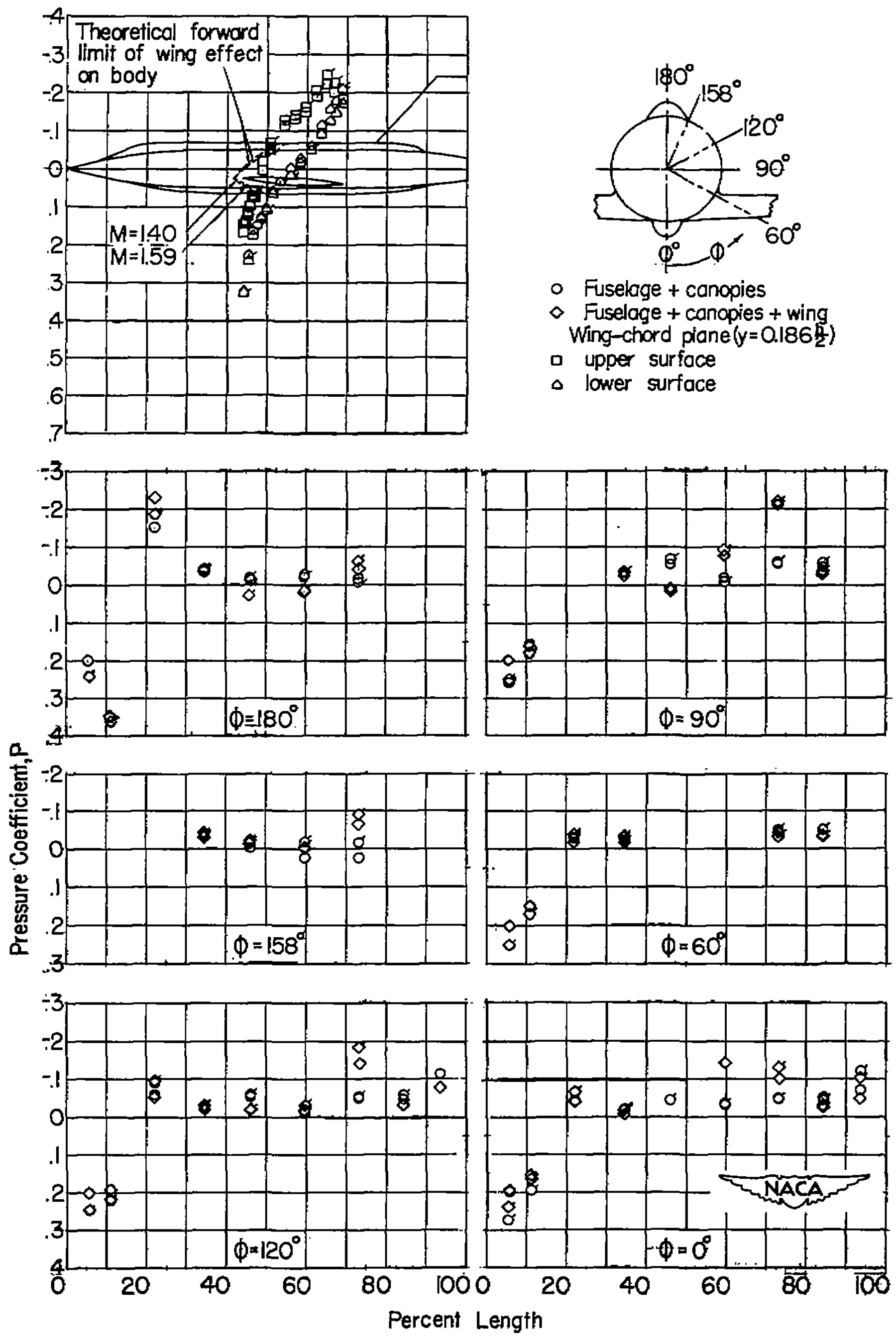



Figure 5.- Normal-force loading distribution as a function of semispan. M = 1.59.




(a) $\alpha = -5^{\circ}$.

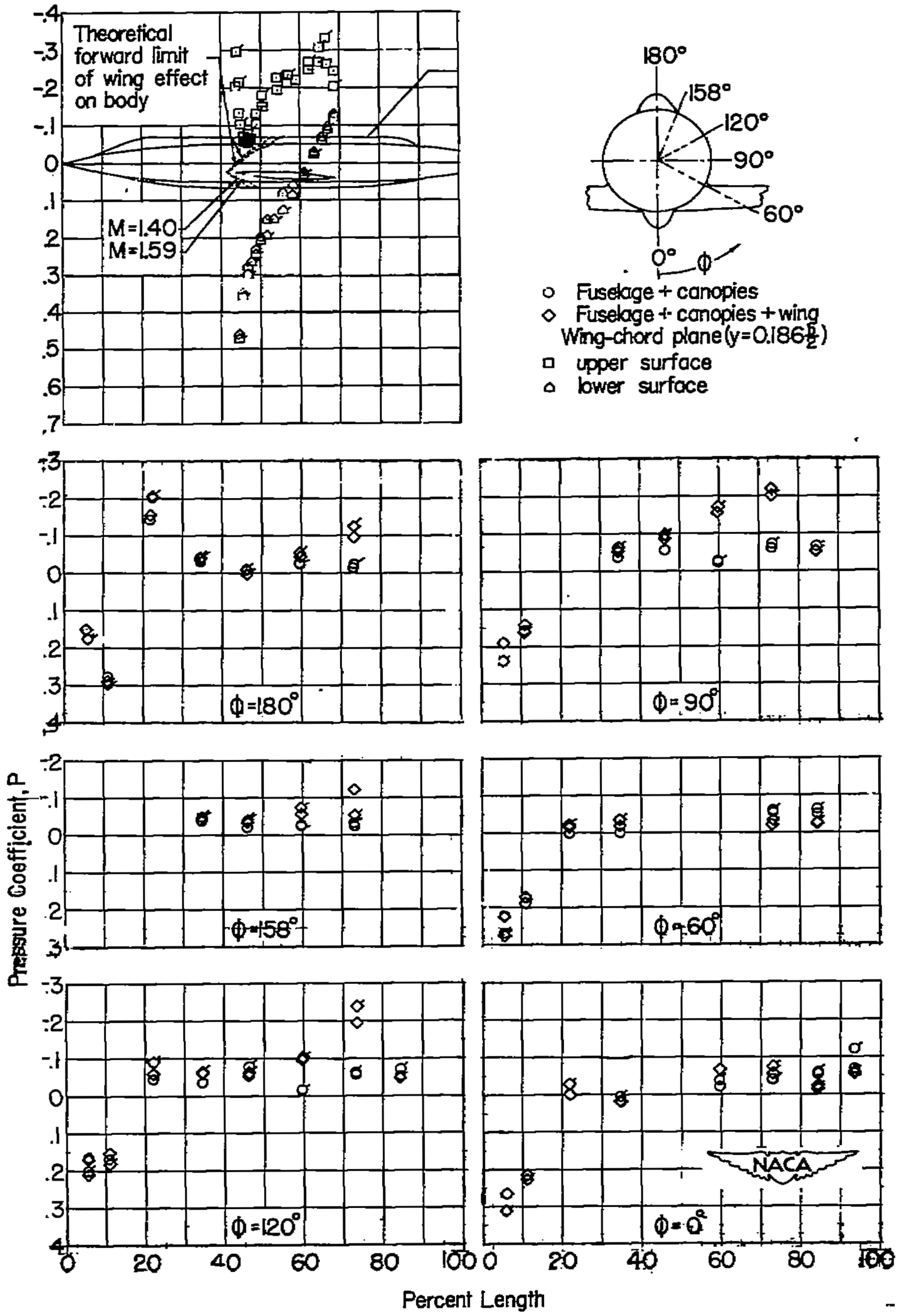
Figure 6.- Pressure distribution over the fuselage and canopies with and without the wing for Mach numbers of 1.40 and 1.59. The flagged symbols denote values at $M = 1.40$.




(b) $\alpha = 0^{\circ}$.

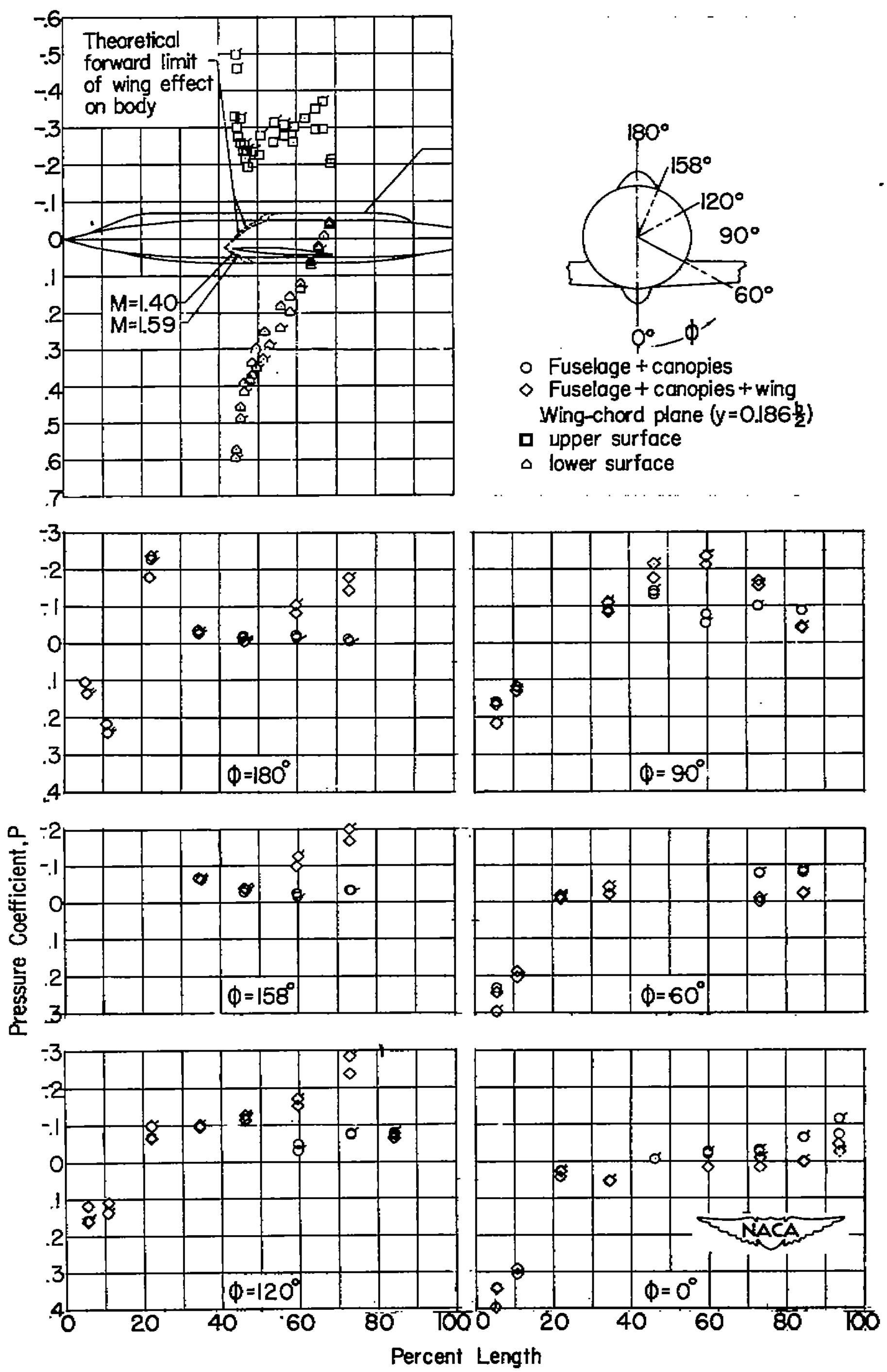
Figure 6.- Continued.

[redacted]



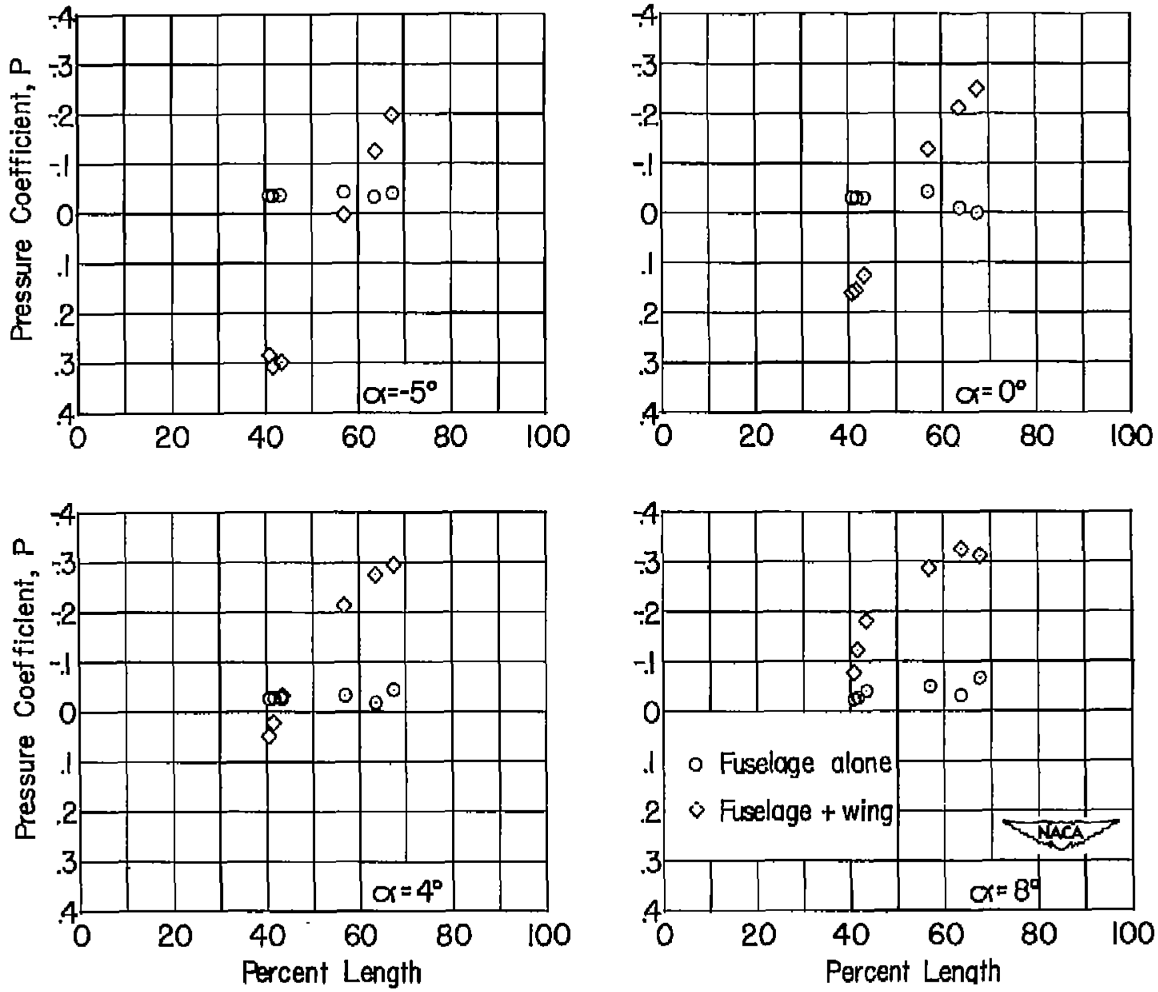

(c) $\alpha = 4^{\circ}$.

Figure 6.- Continued.




(d) $\alpha = 8^{\circ}$.

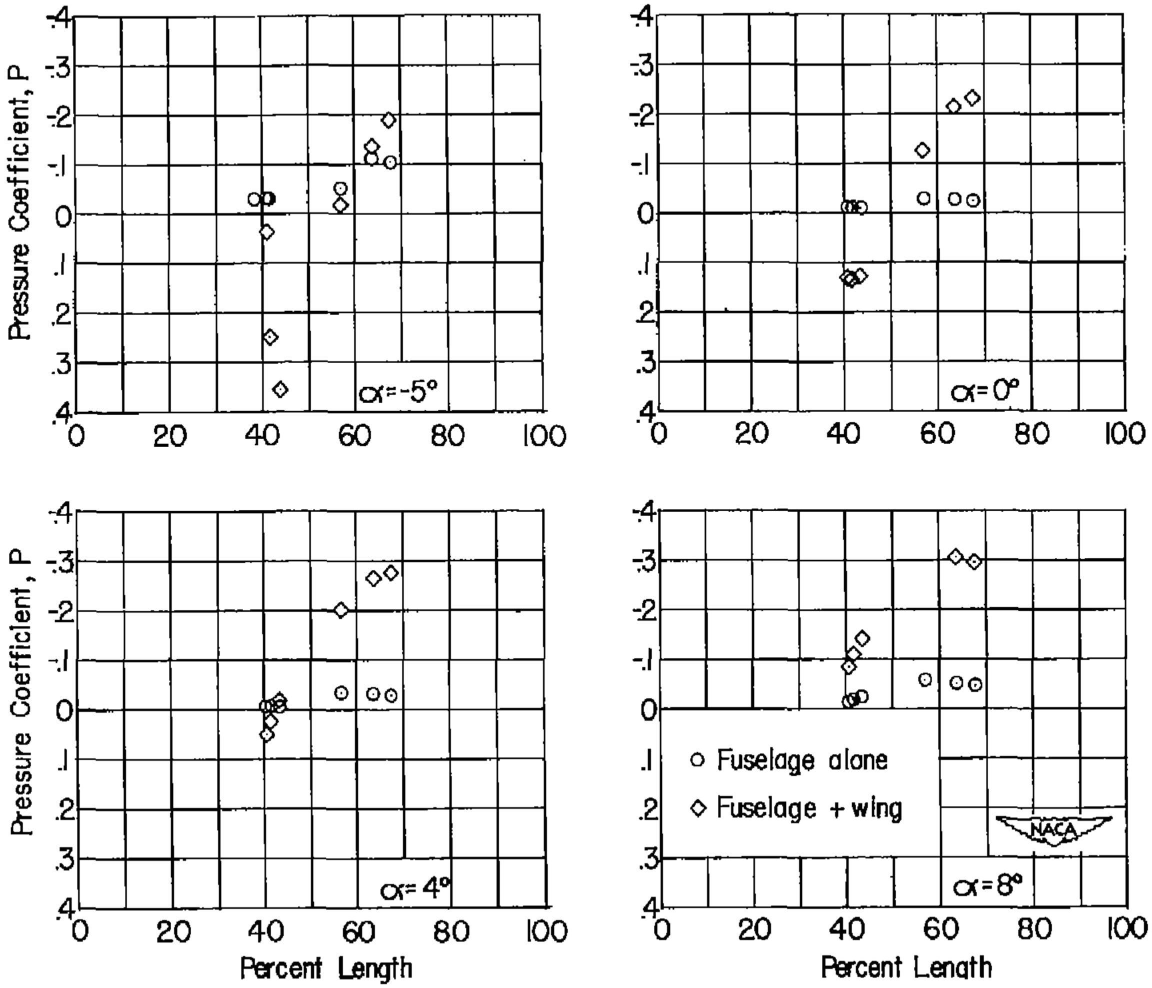
Figure 6.- Concluded.




(a) M = 1.40.

Figure 7.- Pressure distribution over the fuselage at the wing-fuselage intersection for various angles of attack.

[redacted]




(b) M = 1.59.

Figure 7.--Concluded.

[redacted]

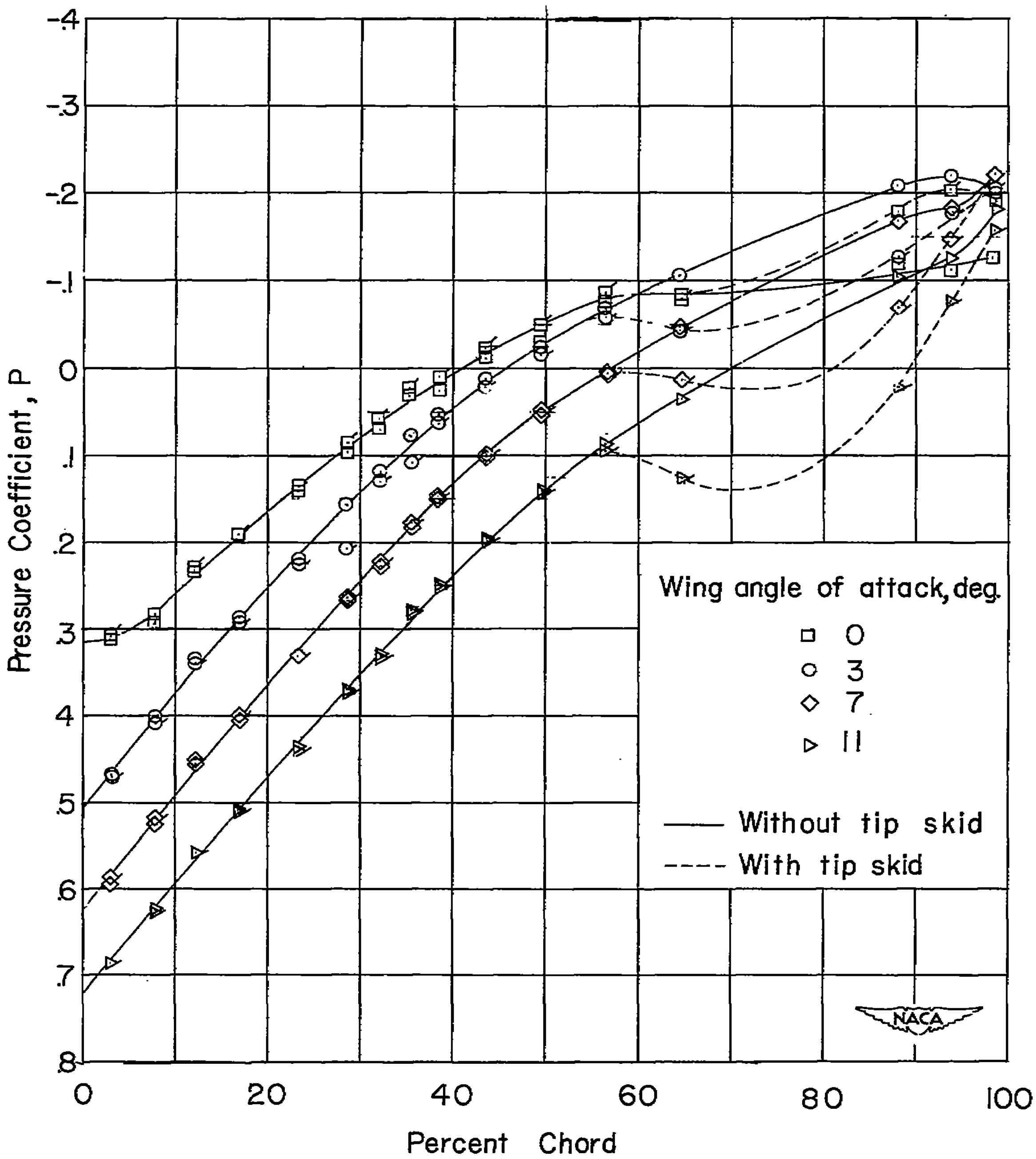



Figure 8.- Pressure distribution over the lower surface of the wing at the 0.937b/2 station with (flagged symbols) and without tip skids. M = 1.59.



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